

A PROPOSED USE OF ACCELEROMETER DATA FOR AUTONOMOUS AEROBRAKING AT MARS

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For any interplanetary mission, there are certain types of data that are used as a means of determining both the position and velocity of a spacecraft. NASA's Deep Space Network (DSN) is employed for the purpose of transmitting and receiving data to and from the spacecraft, respectively. For this exchange of information to take place, both the DSN and spacecraft antennae must be pointed towards each other. This mutual geometry is not maintained throughout aerobraking, specifically while the spacecraft is within the atmosphere. Some spacecraft are equipped with an Inertial Measurement Unit (IMU), which typically is comprised of gyroscopes and accelerometers. The accelerometers provide data pertaining to the spacecraft's acceleration. Since the spacecraft loses telemetry during the drag pass, this research focuses upon the possibility of using accelerometer data as a means of augmenting the current state knowledge of the spacecraft. This knowledge could also assist in obtaining best estimates for subsequent periape times and altitudes (integral aspect of aerobraking operations). The end goal is to achieve some level of autonomy concerning aerobraking operations. To this end, an improved navigation capability is needed. Part of this capability may result from the use of accelerometer data. This paper addresses the road map by which this goal may be implemented; a methodology is given by which an improvement to spacecraft navigation accuracy may be achieved by including accelerometer measurements as observations.

1. INTRODUCTION

For any interplanetary mission, there are certain types of data that are used as a means of determining both the position and velocity of a spacecraft. The data types currently in use are Doppler, Range, Optical, and Δ VLBI. All of them are radiometric with the exception of the Optical data type. NASA's Deep Space Network (DSN) is employed for the purpose of transmitting and receiving data to and from the spacecraft, respectively. For this exchange of information to take place, both the DSN and spacecraft antennae must be pointed towards each other.

To decrease propulsive expenses for a given mission, a spacecraft may be initially placed into a highly eccentric orbit, with periape located within a planet's atmospheric influence. Outside this atmospheric influence, this highly eccentric orbit would theoretically remain unchanged in size due to the spacecraft's presence in a conservative field (gravitational). However, because of the viscous effects that the spacecraft experiences within the atmosphere, the presence of a non-conservative force is

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encountered, commonly known as “drag”. Each “pass” through the atmosphere (drag pass), the spacecraft’s orbit will decrease in size and with it, its period. This decrease is allowed to take place until such time at which operations deems apoapse to have reached its desired altitude. Then, the spacecraft performs a periapse-raising maneuver and the orbit is essentially circularized or is placed in its final configuration. This method of orbit size reduction is commonly known as “aerobraking”. Aerobraking can be viewed as a means to achieve a change in velocity provided *gratis* by the atmosphere, which would otherwise have to be provided by the spacecraft thrusters.

Structurally and thermally, the spacecraft employing this aerobraking method has a preferred orientation for its intended trajectory through the atmosphere. Thus, the spacecraft must slew into this orientation prior to each drag pass. This makes the transmission and reception of data during the drag pass impossible; consequently, there is a gap in the data observables. Transmission is resumed once the spacecraft exits the atmosphere and can re-acquire the Earth with its antenna.

Within the scheme of orbit determination, another caveat to this aerobraking method is that the spacecraft performs its slewing maneuver by command (not autonomously). Since the period changes during each orbit, the amount by which the spacecraft must slew for each consecutive drag pass must be computed by personnel on the ground, after which it is sent to the spacecraft in sequencing commands. As the period decreases, there is less time to perform these computations (ref. 8). This problem is coupled with a light-time travel communications constraint.

Some spacecraft have an Inertial Measurement Unit (IMU), which typically is comprised of gyroscopes and accelerometers. The gyroscopes provide data pertaining to the spacecraft’s attitude, while the accelerometers provide data pertaining to the spacecraft’s acceleration. Since the spacecraft loses telemetry during the drag pass, it is the motivation of this research to indulge in the possibility of using accelerometer data as a means of augmenting the current state knowledge of the spacecraft.

In order to insure spacecraft survivability, aerobraking operations is concerned with being able to predict the subsequent periapse time and altitude. The subsequent periapse time ties into the slewing maneuver execution time. The subsequent periapse altitude ties into the possible need for a periapse-raising maneuver if the altitude will be too low (the main concern is to fly a given design dynamic pressure or heating rate corridor). Accelerometer data alone will not provide insight into the next periapse altitude, because the accelerometer will measure drag without discrimination as to whether that drag is due to an expanding atmosphere or a lower altitude pass. However, the orbit determination process as a whole may avail itself of the accelerometer data in order to assist in quantifying the aforementioned caveat.

Therefore, in order to make the best use of the accelerometer data, the spacecraft must process the orbit determination on-board. Future missions (including the 2005 Mars Reconnaissance Orbiter) will have an on-board payload called “Electra” in order to support a network infrastructure at Mars. If orbit determination software is already on-

board for this support reason, making use of the same to include accelerometer data processing should not cause any impact.

2. METHODS AND PROCEDURES

The ideal goal of this research is to incorporate accelerometer data as an observation type within the scheme of spacecraft navigation, and with this inclusion obtain an improvement in the knowledge of the spacecraft's current/predicted position and velocity. The road map to accomplish this is as follows:

- a) As a post-mission process, obtain accelerometer data from the Mars Global Surveyor (MGS) aerobraking phase and incorporate these accelerations into the trajectory prediction scheme as the drag pass dynamics (i.e. in lieu of using atmospheric models such as the drag equation). Assess the capability of the accelerometer data in reproducing realistic atmospheric effects. This is currently being done.
- b) Obtain accelerometer data from the Mars 2001 Odyssey aerobraking phase as it occurs, and repeat the process described previously. From the trajectory obtained, extract the subsequent periapse time and altitude predictions. Assess the capability of the accelerometer data in predicting the subsequent peripase time to within 5 minutes and altitude to within 10 kilometers. As will be explained further on, these quantities are critical to aerobraking.
- c) Formulate the accelerometer data as a data type (i.e. such as Range and Doppler) and process data (post-mission) from MGS and Odyssey. Assess the capability of the accelerometer data in achieving a better estimate of the state (and/or an improvement over using the accelerometer data as the dynamics).
- d) During the mission, use and incorporate the accelerometer data as the dynamics onboard Mars 2005 Reconnaissance Orbiter (MRO) within the Electra payload. Process accelerometer data as an observation, on the ground.
- e) Provide the capability for future missions (2007 and beyond) to use accelerometer data as a data type, for onboard navigation operations.

Thus far, accelerometer data from the MGS aerobraking phase has been obtained and incorporated into the trajectory prediction scheme as the drag pass dynamics. The capability of the accelerometer data in reproducing realistic atmospheric effects was assessed. To do this, the accelerometer data was formatted into the small forces file format and read into the Orbit Determination Program (ODP) as such. The drag model was "turned off" when generating the trajectory. A reconstructed MGS orbit was then compared to that obtained from using the accelerometer data. The difference between the two gave insight into the capability of the accelerometer in being able to reproduce the atmospheric dynamics experienced by the spacecraft.

For Odyssey, a tool will be created and used to search the telemetry delivery directory for the latest telemetry file. The most recent accelerometer data will then be formatted into a small forces file, and read into ODP. A set of initial conditions will be obtained from the navigation team, and the ODP atmospheric model will be "turned off". A

trajectory using the accelerometer data as the atmosphere will be generated. From this trajectory, a prediction of periapse time and altitude will be extracted. This periapse time and altitude will be compared to the predictions (and/or reconstructions) obtained from the navigation team. The agreement between the two should give insight into the accelerometer's capability of predicting quantities that are critical to aerobraking.

During the aerobraking phase periapse time is critical because this is when the spacecraft is deepest within the atmosphere. Therefore, aerobraking event sequences are referenced to periapse time. The combination of the attitude control deadband, the rotation rate of the spacecraft attitude reference frame, and the angle through which the spacecraft rotates during a drag pass, imposes the time-from-periapse constraint for the spacecraft to get into and out of its aerobraking attitude configuration. This is the importance of being able to predict the periapse time to within the historical 5 minute timing margin.

Another critical aspect of aerobraking is dynamic pressure or heat rate. The spacecraft trajectory is designed to fly in a given dynamic pressure or heat rate environment. This environment has an upper and lower bound. If the atmospheric viscous effects are too high, then the mission may be physically lost due to exceeding thermal/structural tolerances. If the atmospheric viscous effects are too low, then the mission may fail to meet a final orbit constraint at the designed time (i.e. mean local solar time descending node crossing). Therefore, it is said that the spacecraft is to "fly" in a given dynamic pressure or heat rate corridor. To keep the spacecraft within this designed corridor is called "corridor control" (ref. 8).

The accelerometer data (along with initial drag pass conditions) may be used to predict the subsequent periapse time. However, accelerometer data alone cannot predict the subsequent periapse altitude. The viscous effects, which the accelerometer measures, may be due to the spacecraft's flying one of two different scenarios:

- a) Flying in an expanding atmosphere (i.e. dust storm activity) at an expected altitude
- b) Flying at a lower altitude than expected

The prediction scheme must be able to differentiate one from the other. This capability is critical to mission survivability because periapse altitude variation may be tolerable, but not an expanding atmosphere. Therefore, astrodynamics must be used in conjunction with the accelerometer data, in order to provide for this capability. Atmospheric densities are referenced to a given altitude. So, at a fixed altitude the density should not be increasing exponentially. If this behavior seems to take place, then it is a good identifier for a possible expanding atmosphere scenario.

The next phase of research consists of formulating the accelerometer data as an observation type. Thus far, accelerometer data is used as the non-conservative dynamics. Typical uses for accelerometer data (for interplanetary missions) have been to quantify maneuvers (i.e. Trajectory Correction Maneuvers {TCMs}), orbit insertions, and

atmospheric entries (i.e. Entry-Descent and Landing {EDL} operations), to name a few. Earth based missions (such as GRACE), will use the accelerometer data as a means of measuring non-gravitational accelerations to subtract them from the dynamics in order to refine the Earth's gravitational field. The suggested use of accelerometer data as a measurement has been suggested as early as 30 years ago. However, this method of using the accelerometer data has yet to be implemented for various reasons. One of them is that accelerometer accuracies were not as good as they are now. Accelerometer noise characteristics are the same now, but more controllable than in the past. With improvements in inertial sensor technology, it may be very feasible to return to the suggested use of accelerometer data as a measurement type.

Therefore, this phase of research proposes to formulate the accelerometer data type, and implement its use within orbit determination. Both MGS and Odyssey aerobraking data will be used as a means to assess the accelerometer data type as a viable measurement source. Navigation team solutions will serve as the comparative baseline. A comparative methodology that will be used concerns use of the accelerometer data as the dynamics (as previously described) versus as a measurement.

The next phase on the road map consists of processing the accelerometer data as both the dynamics (onboard MRO) and as an observation (on the ground). To process the accelerometer data onboard, a navigation capability would be needed onboard. There is also the need for a CPU to use for this purpose. What has been proposed for MRO (and missions beyond) is a payload called "Electra". This payload has the capability of processing two-way and *in situ* one-way Doppler data. The motivation for this is the design and implementation of a robust navigation/communication infrastructure at Mars (Mars Network). Therefore, it would be a matter of taking advantage of an existing design and simply adding an augmented capability to it. Adding this capability should be of no consequence or complicating impact. It is also possible that Electra could process accelerometer data as a standalone data type. Research in this area will uncover the feasibility, advantages, and disadvantages of such a scenario.

Future missions may use the Electra payload in a multiple observation type environment. It is feasible to envision a future mission to perform onboard navigation with an already orbiting asset at Mars (i.e. MRO), and thus be able to formulate the *in situ* data type and use this along with accelerometer data as a means of improving state estimates. This capability would already be integrated within the payload.

As far as error sources are concerned, the following list of elements is typical for accelerometers. Definitions for these elements are found in (ref. 2):

- a) *Bias*
- b) *Scale Factor*
- c) *Sensitivity*
- d) *Non-linearity*
- e) *Repeatability*

The bias can be defined as the accelerometer output with no input acceleration present. Bias is a signed quantity usually expressed in units of acceleration. Before each drag pass, the bias can be evaluated and subtracted off of the measured accelerations. Thus, dealing with the bias is a straightforward procedure. The bias stability (drift) is a function of temperature fluctuations. For MGS, the accelerometer temperature was maintained to within 0.234 degrees Celsius over the entire 201 aerobraking passes of Phase 1. This corresponded to a bias drift of approximately 1% (ref. 3). The following figure shows the raw output of the accelerometer for a given orbit.

Figure 1.

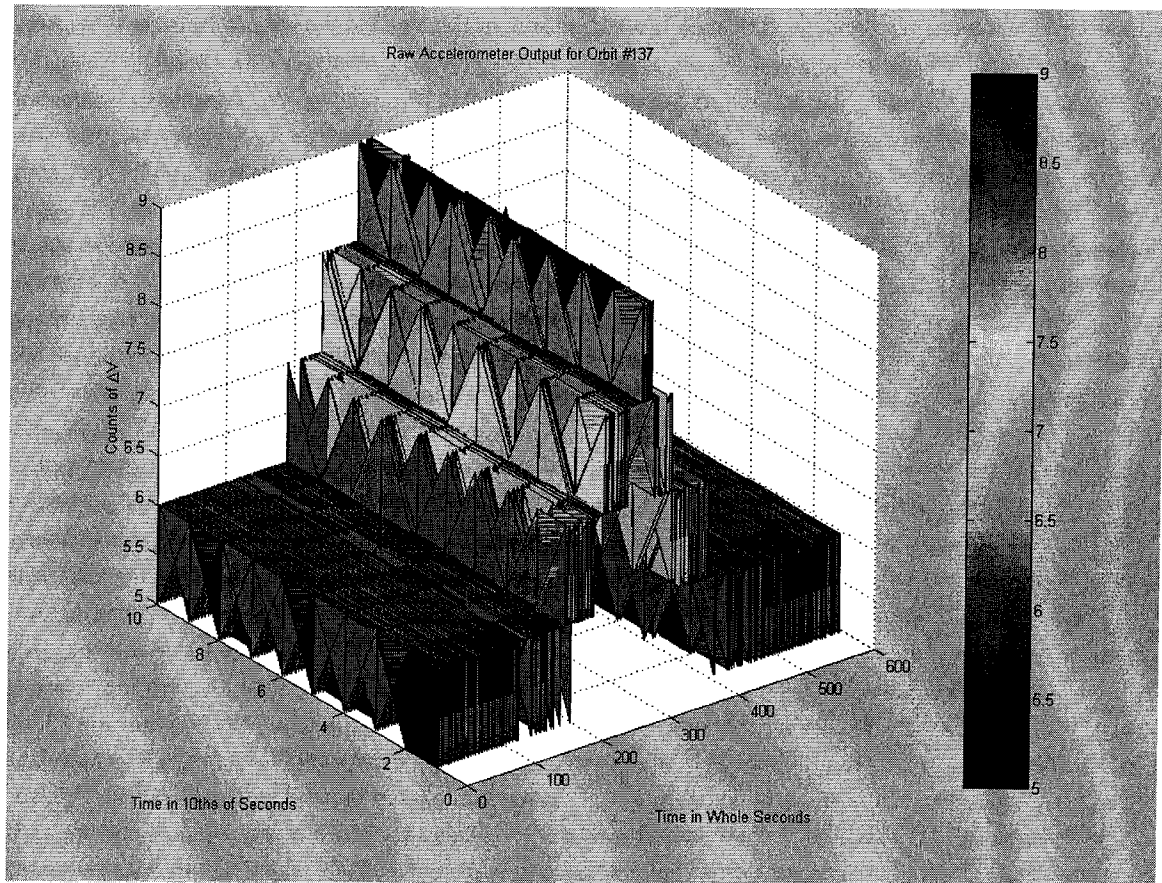


Figure 1. MGS Accelerometer output for orbit number 137

The manner in which to deal with this bias, is to monitor the output in an environment where no output should exist. Then, the mean is taken for the values recorded. This mean value is effectively the accelerometer bias. The bias can then be subtracted off of the measured acceleration experienced during the drag pass. For MGS aerobraking, the bias was seen as an output varying between 5 and 6 counts per measurement, with a mean of approximately 5.7 counts (± 0.005 counts). To make the data more coherent and useful, the counts are integrated (summed) over each second, and then converted into units of

acceleration. At this point, the data is still plagued by noise, so the profile is “smoothed” with a running mean. The following figure shows the acceleration profile produced from this process.

Figure 2.

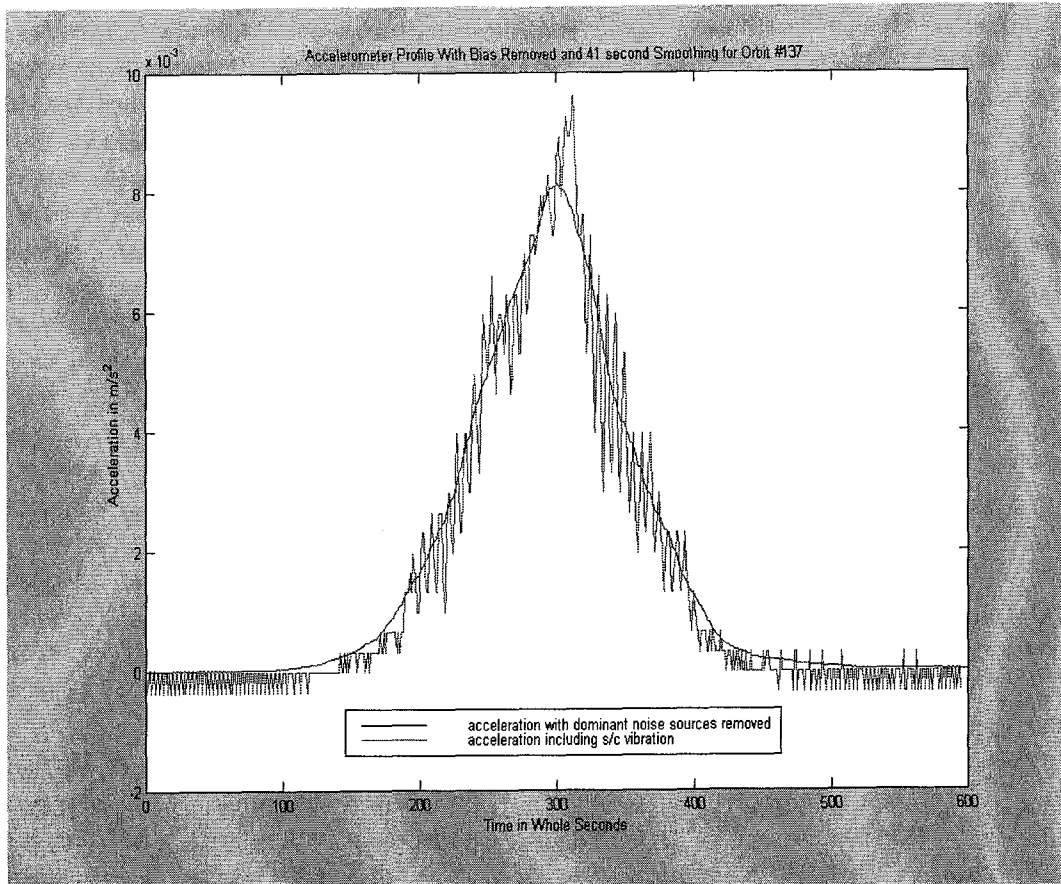


Figure 2. MGS accelerometer profile after processing/converting the raw data into useful data

It must be noted, however, that bias is an inherent property of the accelerometer instrument. Bias fluctuations (or drift of the mean value) are mostly due to variations in instrument temperature. Therefore, the temperature must also be monitored in order to quantify the bias drift.

The scale factor can be defined as the ratio of the change in output (in volts or amperes) to a unit change of the input (in units of acceleration). It is typically given in mA/g or V/g. The scale factor error is expressed in percent or in parts per million (of full scale). The scale factor is sensitive to the same factors as the bias. Thus, the scale factor remained acceptably constant.

The sensitivity can be defined as the ratio of a change in response to a change in an undesirable or secondary input (as the scale factor variation to a unit of power supply voltage change). Next, the non-linearity can be defined as the deviation of the

accelerometer output from the input-output linear fit over the operating range. The deviation is expressed as a percentage of the full-scale output. An optional representation is to provide non-linearity coefficients (i.e. $K2$ ($\mu\text{g/g}$) and $K3$ ($\mu\text{g/g}^3$)). Finally, the repeatability concerns the closeness of agreement among measurements of the same variable, repeated under the same conditions, especially when changes in conditions occur or when operation is interrupted between the measurements.

One quantity that is inherent to sensors, such as accelerometers, is noise. Noise can be interpreted as undesired perturbations in the accelerometer output signal, which are generally uncorrelated with desired or anticipated input accelerations. Noise in the accelerometer output is of two types: intrinsic and seismic.

- a) Intrinsic noise is generated within the accelerometer and represents the limiting factor in making measurements. Intrinsic noise is random in nature and is characterized by a noise power spectral density (PSD) curve. For MGS, the accelerometer used has low intrinsic noise.
- b) Seismic noise is a true input acceleration (usually unanticipated by the user). It results from noise sources in the local environment (running motors, seismic shocks, etc.) and their transmission to the accelerometer through the mounting structure that supports the accelerometer. For MGS, the main source of seismic noise is due to a cracked solar panel. This introduced a 6.7 second periodic signal buried within the output. Subsequent sections will describe this further.

The possible accuracy of range data is typically on the order of 1 m and Doppler data on the order of 1 mm/s (S-band, with 0.1 mm/s X-band). The accuracy of the MGS accelerometer data is on the order of 1 μg . It must be noted, however, that although the accelerometer can measure accelerations on the 1- μg level, one cannot gain insight into them. This is because of the 0.332 mm/s quantized output (possible at each 0.1 second measurement interval)(ref. 3). In effect, this can be translated into a quantized acceleration output of 3.32 mm/s² (or 338 μg for any given measurement made in a 0.1 second interval). Providing for further understanding, this means that if the spacecraft (at a given moment) were to suddenly experience a constant acceleration of 1 μg (assuming it was previously in a $< 1 \mu\text{g}$ environment), it would take 33.8 seconds (from that moment) for the accelerometer to have an output. This is assuming no bias. Therefore, during the first 33.7 (of 33.8) seconds, the accelerometer would have no output. This is a caveat of the quantization process. However, this output resolution is acceptable if the acceleration data sources of interest are thrusting maneuvers and accelerations due to drag (ref. 4).

The acceleration measured by the accelerometer can be thought of as being comprised by the following elements:

- a) *Drag*
- b) *Gravity-gradient*
- c) *Bias*
- d) *Vibration*

- e) *Angular motion*
- f) *Solar Radiation Pressure*
- g) *Thrusting (orbit or attitude maneuvers)*

Now, out of the previous elements that comprise the measured acceleration, only the following will be considered:

- a) *Drag*
- b) *Thrust (only those for orbit maneuvers)*
- c) *Angular Motion (could be caused by attitude maneuvers)*
- d) *Bias*
- e) *Vibration*

The other elements that could comprise the measured acceleration cannot be measured by this particular accelerometer (i.e. accelerations $< 1 \mu\text{g}$). Therefore, they have been neglected. Out of the elements that will be considered, several of them are unwanted for orbit determination purposes. The unwanted elements are the following:

- a) *Angular Motion*
- b) *Bias*
- c) *Vibration*

Because these elements can be measured by the accelerometer, they must be subtracted from the measured acceleration. Therefore, the elements used for the observed acceleration are those due to drag and thrust (orbital maneuvers).

The following example will be used in order to quantify the typical acceleration contributions given by each element comprising the measured acceleration. First, assume a 1-sol (duration of one Martian sidereal day) orbit about Mars with a periapse altitude of 100 kilometers. For the sake of simplicity, assume a critically inclined retrograde orbit with a longitude of ascending node and argument of periapse of both zero degrees. Therefore, we have the following classical orbital elements at periapse (all values given in km or degrees):

<i>a</i>	<i>e</i>	<i>i</i>	ω	Ω	<i>v</i>
20415	0.828842	116.6	0.0	0.0	0.0

For this scenario, the contributions to the measured acceleration that can be typically expected are approximately as follows (all values given in m/s^2):

<i>Drag</i>	<i>Gravity Gradient</i>	<i>Thrust (given in g's)</i>	<i>Angular Motion</i>	<i>Solar Pressure Radiation</i>
3.01E-2	1.83E-6	1 μg -25 g	8.0E-4	1.0E-8

From the previous table, the thrust term given is the actual measurement range of the accelerometer. Therefore, the thrust can be anywhere within this range. As the spacecraft

moves further away from the central body, the gravity gradient term decreases approximately as the inverse cubed of the distance. As it is, the accelerometer cannot even measure the gravity gradient acceleration. However, there may be a situation where this term could be measured (i.e. very low altitude periaapse and a highly eccentric orbit). Bias and spacecraft vibration must be removed and filtered respectively (as seen in figure 2). It is also clear that the solar radiation term can be neglected. The angular motion term must be computed, then removed from the measured acceleration since it is not of interest for orbit determination. The following figure is an example of the acceleration profile

3. RESULTS

There are no final results thus far. Although intermediate results have been previously mentioned or described, one goal of the research was to ascertain the reliability of the accelerometer output. To this end, a simulation was performed and compared against the accelerometer data. The simulation consisted of obtaining the MGS navigation team solution for a given orbit, and taking the same orbit while replacing the drag model for the accelerometer output. The difference at the subsequent apoapse (following the drag pass) between the two orbits was only 1.7 km (for a 27000 km apoapse radius). Therefore, the accelerometer was able to faithfully reproduce the general dynamic trends experienced by the satellite during its drag pass.

4. CONCLUSIONS AND RECOMMENDATIONS

While aerobraking, telemetry is typically lost during the drag pass. Being that accelerometers are part of IMUs (which are part of typical payloads), there may be advantages in using the accelerometer data to aid in navigation capabilities. There are aspects of the aerobraking process that could benefit from being automated (and a must, for outer planets). A road map describing the steps by which this may become possible, has been established. Thus far, there have been positive indicators that accelerometer data may be useful in enhancing the spacecraft navigation scheme. This paper is merely a brief introduction into the topic, and sets the stage for further related papers that will develop as a natural process of following the road map previously described.

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